RESULTS OF AN AEROELASTIC TAILORING STUDY FOR A COMPOSITE TILTROTOR WING

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ABSTRACT

The feasibility of a composite tailored wing for a high-speed civil tiltrotor is addressed using existing analytical methods. Composite tailoring is utilized to increase the proprotor aeroelastic stability margins for a thin wing (18% *t*/*c*) designed to improve high speed performance and productivity. Structural tailoring concepts are applied to the wing alone to improve the stability of the symmetric wing beamwise bending mode and the symmetric wing chordwise bending mode, which are the two most critical modes of instability. Skin laminate tailoring is shown to favorably influence the wing pitch/bending coupling and improve the stability of the wing beamwise mode. The wing chordwise mode stability is reduced by skin laminate tailoring due to a decrease in wing stiffness, but by tailoring the distribution of stringer and spar cap areas, the wing chord mode stability can be recovered. Parametric studies show that the overall stability gains from composite tailoring can be limited because of conflicting structural design requirements imposed by the two critical modes of instability, and the necessity to balance the stability boundaries for both modes. The parametric studies are used to define an 18% *t*/*c* tailored wing configuration that meets the stability goals with a minimum weight penalty.

INTRODUCTION

One of the principal design challenges for a high-speed tiltrotor is achieving acceptable proprotor aeroelastic stability margins in airplane mode. Proprotor aeroelastic instability, frequently referred to as proprotor whirl flutter, occurs in the high-speed airplane mode flight regime and can restrict the operating airspeed of a tiltrotor. The instability is caused by destabilizing rotor forces that are generated by the rotor flapping response to the bending and pitching motions of the low-frequency wing modes. This aeroelastic phenomenon has been addressed by industry, university, and government investigators through analyses, model tests, and full-scale flight test programs. In Ref. 1, model tests and analysis of several tiltrotor configurations are described which formed the basis for later tiltrotor configuration development. Both the analytical and experimental results showed the important role of wing and pylon frequency placement to prevent proprotor instability. Ref. 2 describes V-22 aeroelastic model stability tests and analysis correlation in which the wing chordwise stiffness was shown to significantly impact stability boundaries. The sensitivity of stability boundaries to a systematic variation of all wing stiffness components is presented for a conventional untailored wing configuration in Ref. 3.

As discussed in the above literature, proprotor stability is influenced by the rotor, airframe, drive system, and control system; thus, a new design must account for the interaction of all of these elements to ensure acceptable stability margins. When considering only the airframe contribution to proprotor stability, the two most important factors affecting proprotor stability are the frequencies and mode shapes of the wing bending and torsion modes. In fact, the wing stiffness requirements for proper frequency placement to prevent proprotor instability are typically as demanding as the wing strength requirements. The stability boundary is sensitive to the pitch/bending coupling in the wing mode shapes,

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and this coupling can be controlled by relative frequency placement of the wing modes, offset of the pylon center of gravity relative to the wing elastic axis, and structural bending/torsion coupling of the wing torque box.

To meet stability requirements, conventional tiltrotor wing designs use thick wings (23% *t*/*c*) that efficiently provide high torsional stiffness at minimum weight. To improve tiltrotor high speed performance and productivity, it is desirable to reduce the wing thickness ratio (*t*/*c*) without increasing the weight. Performance analyses show that reducing the wing thickness to 18% *t*/*c* decreases the airframe drag by 10% and provides a substantial improvement in aircraft productivity. For a conventional tiltrotor wing design, reducing the wing thickness ratio also decreases the stability boundary due to the loss in stiffness. The stability boundary can be recovered by adding stiffness to restore the mode shapes and frequency placement; however, the additional weight reduces aircraft productivity.

Composite tailoring provides an opportunity to increase the stability of tiltrotors with thin wings without incurring a large weight penalty by favorably modifying the mode shapes and frequency placement of the fundamental wing modes. Composite tailoring has been successfully applied to the design of the swept forward wing of the high performance X29 fixed-wing aircraft to improve flutter and divergence boundaries, and has been analytically investigated as a means to improve the aeromechanical stability of helicopters in Ref. 4. Recent advances in composite manufacturing technology have resulted in lower cost, higher strength design concepts based on composite rod technology which can be applied to tiltrotor wings as described in Ref. 5. The rod manufacturing technology is compatible with composite tailoring and promises to reduce the cost and weight of future composite tailored tiltrotor wing designs. In Ref. 6, tests were performed on composite panels fabricated with unbalanced skin laminates and skewed stiffeners. These tests showed that tailored skin laminates could alter the extension/shear coupling in the panel with the potential for controlling the structural bending/torsion response of a wing.

In this paper, composite tailoring is applied to the design of a thin wing (18% *t*/*c*) for a civil tiltrotor which cruises at 375 knots true airspeed (KTAS) at 20,000 ft altitude. The motivation for the current study is the need to improve high-speed tiltrotor performance and productivity by developing thinner, lighter wings. The goal of the composite tailored wing is to achieve the same or higher level of stability as the 23% *t*/*c* V-22 tiltrotor wing, which serves as the baseline configuration. The study addresses the effects on stability of the wing skin and spar web laminate composition, stringer and spar cap area distribution, and wing thickness ratio. Parametric studies of each of these design variables provide a basis for the design of a composite tailored 18% *t*/*c* wing which satisfies the proprotor stability goals with minimum weight impact.

COMPOSITE TAILORED WING DESIGN APPROACH

Baseline Configuration Definition

Accurate prediction of aeroelastic stability boundaries requires detailed mathematical models for the rotor, drive system, and airframe. The most realistic assessment of the benefits of composite tailoring is made with math models representative of an actual design, in which the rotor, fuselage, wing, and pylon structural parameters are fully developed and known accurately. For this reason, the V-22 Full-Scale Development (FSD) configuration shown in Fig. 1 was chosen as the baseline configuration for the tailoring study. The V-22 carbon-epoxy composite wing provides an excellent reference configuration for this study because the V-22 is a mature design with correlated rotor and airframe dynamic models, confirmed weights data, available structural loading conditions, and verified stability results. The V-22 rotor, drive system, pylon, engine, fuselage, and empennage are used without modification in this study and only the wing is tailored. Furthermore, the basãeline V-22 wing planform, airfoil, sweep, and taper are maintained throughout this study to isolate the benefits of composite tailoring from those associated with geometric configuration changes.

Tailored Wing Design Criteria

The proposed stability design envelope for a 40-passenger civil tiltrotor designed to cruise at 375 KTAS at 20,000 ft is presented in Fig. 2, which also shows the predicted stability boundary for the baseline V-22 configuration. Although the V-22 has insufficient installed power to cruise at 375 KTAS at 20,000 ft, the predicted V-22 stability boundary exceeds the civil tilt-rotor requirements. The design goal for the tailored wing study is to meet or exceed the V-22 stability boundary, and therefore exceed the minimum stability specifications for the proposed

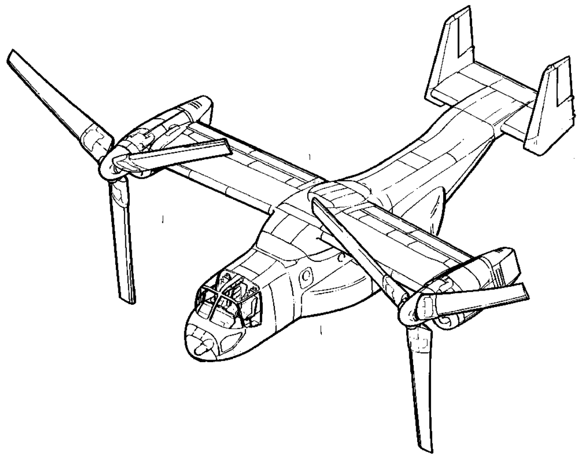


Fig. 1. Baseline V-22 tiltrotor configuration.

civil tiltrotor. Additional wing weight could be saved by designing the tailored wing to exactly meet the civil tiltrotor stability requirement, but this approach would not permit direct comparison to the V-22 reference configuration to quantify the improvements due to structural tailoring. All stability comparisons in this study are made for the aircraft operating at a density altitude of 7,500 ft, with the pylons locked to the downstop, and the rotor rpm at 84%.

Proprotor Stability Design Drivers

The physical mechanism of proprotor instability is discussed in detail in Refs. 1 and 2 and will be briefly reviewed here. Tiltrotors can exhibit proprotor instability in the fundamental wing bending modes in the high speed airplane mode flight condition. Historically, the most critical modes have been the symmetric wing beamwise bending mode and the symmetric chordwise bending mode. For these modes, the rotor can create destabilizing inplane hub forces which can overcome the structural and aerodynamic damping of the wing at high speed and cause an instability. In high speed airplane mode, the rotor generates inplane shear forces in response to the rotor mast perturbation pitch angle and pitch rate. From a stability viewpoint, the mast pitch angle and pitch rate are dictated by the response of the low-frequency wing modes, such as the symmetric wing beamwise bending (SWB) mode, symmetric wing chordwise bending (SWC) mode, and symmetric wing torsion (SWT) mode. The rotor inplane forces originate from blade perturbation lift forces which are resolved into the inplane direction due to the very large inflow angle in airplane mode. The magnitude

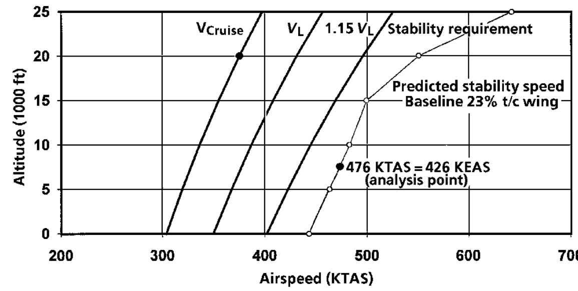


Fig. 2. Civil tiltrotor airspeed/altitude envelope.

and phase of the inplane hub forces are governed by the airspeed as well as amplitude and frequency of the mast motion. Proprotor stability can be influenced by changing the frequency placement of the wing modes and by modifying the mode shapes to alter the mast dynamic response and reduce the destabilizing inplane hub forces.

Composite wing tailoring can be used to minimize the mast pitch motion in the fundamental wing modes affecting proprotor stability. Fig. 3 illustrates that by minimizing the mast pitch component, the rotor destabilizing forces are reduced and the stability boundary is increased. In a typical tiltrotor wing design, the rotor mast pitches nose up (θ_Y) as the wing bends upward (Z) in the wing beam bending mode. The noseup pitch is caused by inertial forces and moments due to the rotor/pylon mass offset from the wing shear center and the proximity of the wing bending and torsion frequencies. For a conventional composite wing design with structurally balanced skin laminates, the wing provides no structural pitch/bending coupling to resist the noseup pitch due to pylon mass offsets. Unbalanced composite skin laminates, on the other hand, can create nosedown structural twist as the wing bends upward to offset the pitchup tendency from the pylon mass offsets. The net effect is reduced pitch/bending coupling and improved stability.

The preceding discussion applies to the SWB mode, which is typically the most critical mode of instability. However, as wing skin tailoring schemes are employed to improve the stability of the SWB mode, the SWC mode can become the most critical and limit the flight envelope of the aircraft. The SWC mode is not responsive to improvements in SWB pitch/bending coupling and, in fact, the stability is decreased by skin tailoring due to the reduction in stiffness. The stability of the SWC mode is improved by increasing the chordwise

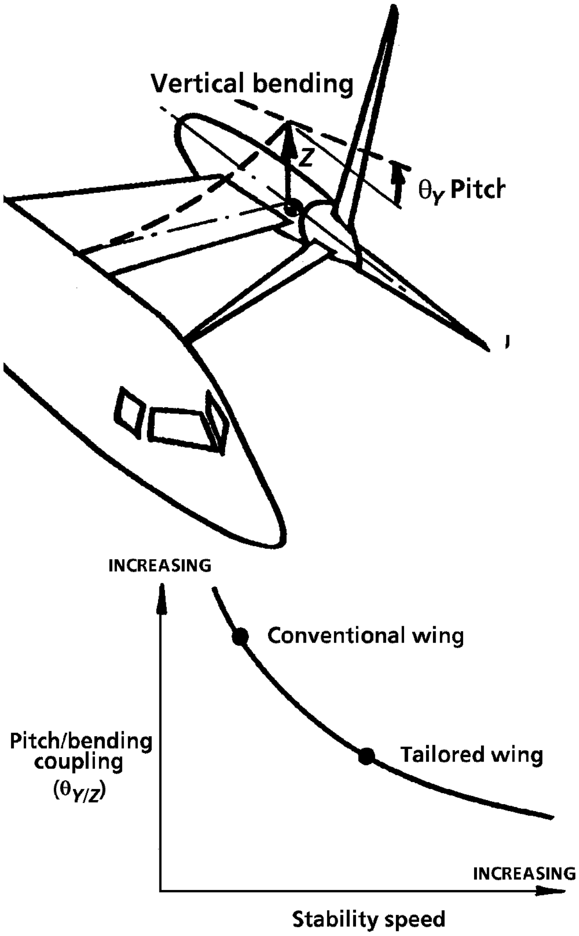


Fig. 3. Composite tailoring influence on pitch/bending coupling and prop-rotor stability.

bending stiffness by tailoring the cross sectional distribution of stringer and spar cap areas.

TAILORED WING CONFIGURATION ANALYSIS METHODOLOGY

Analysis Flow Diagram

The composite tailored wing configuration analysis involves several computer codes which are linked together as illustrated in the flow chart in Fig. 4. Following the initial selection of the skin laminate composition, a Bell-developed preprocessor, CPS04P, is used to generate skin laminate properties for input to the NASTRAN FEM of the tailored wing. The NASTRAN models of the tiltrotor are then used to generate the mode shapes and frequencies required by the Aeroelastic Stability Analysis of Proprotors (ASAP). For configurations which meet the stability criteria, a strength assessment is made using the Bell Loads Development and Analysis Methodology (LODAM) and the Computer-Aided Stress Analysis (CASA) system, which includes three detailed stress analysis programs. These analyses are the same codes used on the V-22 FSD tiltrotor program. Finally, a weight assessment is made by calculating the weight change due to the addition and removal of structural weight items.

Composite Laminate Modeling Procedure

The analysis of the tailored wing configuration begins with a definition of the skin laminate equivalent properties as shown in Fig. 5. The skin laminate is defined by specifying the ply material properties such as E, G, μ, the number of plies, and the thickness and orientation of each ply in the stacking sequence. For a conventional untailored wing, the skin laminate is a balanced blend composed of approximately the same number of -45-deg plies and +45-deg plies, including several 0-deg and 90-deg plies to provide adequate strength in all directions. A composite tailored laminate is an unbalanced blend with a high proportion of -45-deg plies to +45-deg plies, but retaining the required 0-deg and 90-deg plies for strength.

Individual ply properties and orientations are input to the CPS04P laminate analysis, which computes equivalent properties for use with the NASTRAN CQUAD4 plate element representation used in the NASTRAN model of the skin. The CQUAD4 element is represented by the fully coupled laminate equations shown in Fig. 6, and can account for all coupling effects introduced through unbalanced laminates. Referring to Fig. 6, the governing laminate force and moment equations are expressed in matrix form consisting of a membrane stiffness matrix [A], a membrane/bending coupling stiffness matrix [B], and a bending stiffness matrix [D]. These matrices relate the laminate force [N] and moment [M] resultants to the impressed mid plane strains [ε] and curvatures [K] of the laminate. The [A], [B], and [D] matrices are computed by the CPS04P laminate analysis and are specified on NASTRAN PSHELL and MAT2 cards which are referenced by the CQUAD4 element.

In the membrane stiffness matrix of a perfectly balanced laminate, the A_{16} and A_{26} terms are zero. For typical laminates, these off-diagonal terms are an order of magnitude lower than the diagonal elements

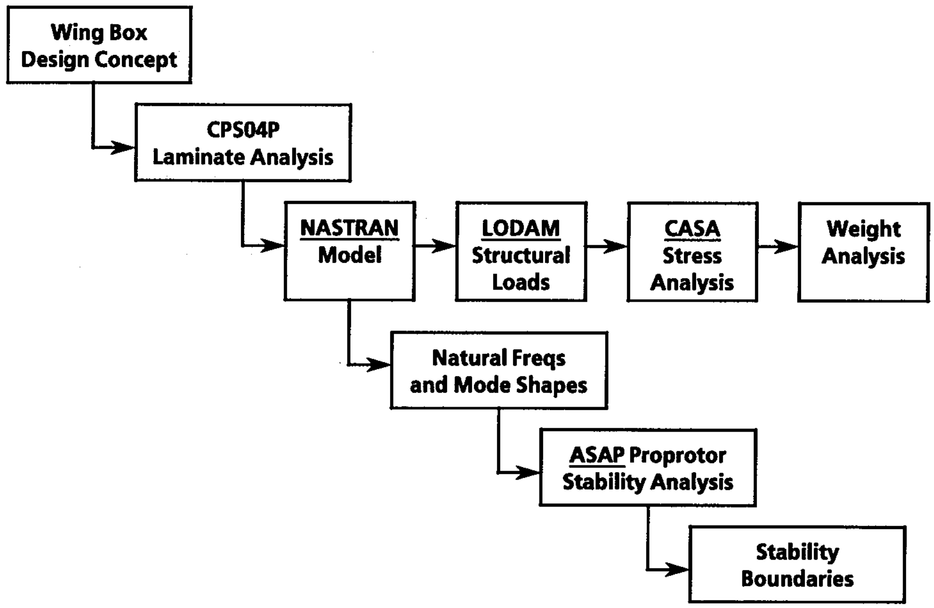


Fig. 4. Tailored wing configuration analytic flow diagram.

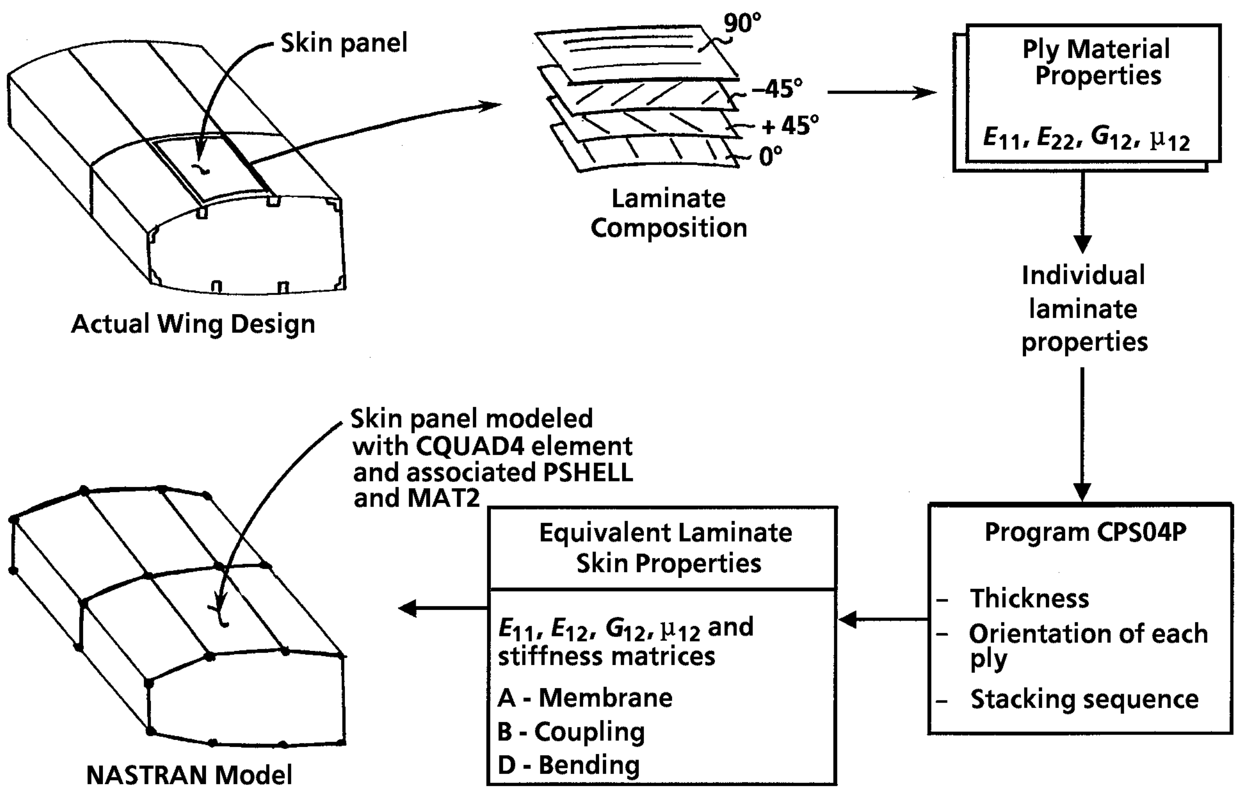


Fig. 5. Procedure used to represent all structural coupling terms for a tailored composite skin laminate.

of the [A] matrix. With an unbalanced laminate, the A_{16} and A_{26} terms are the same order of magnitude as the diagonal terms. The off-diagonal terms generate extension/shear coupling which can be used to modify the pitch/bending coupling in the wing. One of the undesirable results of using unbalanced laminates is a

$$\begin{bmatrix} N_x \\ N_y \\ N_{xy} \end{bmatrix} = \underbrace{\begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{21} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix}}_{\text{Membrane stiffness matrix}} \begin{bmatrix} \varepsilon_x^o \\ \varepsilon_y^o \\ \gamma_{xy}^o \end{bmatrix} + \underbrace{\begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{21} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix}}_{\text{Membrane/Bending coupling matrix}} \begin{bmatrix} K_x \\ K_y \\ K_{xy} \end{bmatrix}$$

$$\begin{bmatrix} M_x \\ M_y \\ M_{xy} \end{bmatrix} = \underbrace{\begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{21} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix}}_{\text{Membrane/Bending coupling matrix}} \begin{bmatrix} \varepsilon_x^o \\ \varepsilon_y^o \\ \gamma_{xy}^o \end{bmatrix} + \underbrace{\begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{21} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix}}_{\text{Bending stiffness matrix}} \begin{bmatrix} K_x \\ K_y \\ K_{xy} \end{bmatrix}$$

Or in short form

$$\left[\frac{N}{M}\right] = \left[\begin{array}{c|c} A & B \\ \hline B & D \end{array}\right] \left[\frac{\varepsilon^o}{K}\right]$$

Fig. 6. General laminate equations used in composite tailored wing analysis in NASTRAN.

decrease in the torsional stiffness of the wing box. The reduction in torsional stiffness is caused by the increase in shearing deformation due to transverse shears from an applied torque. Increased shearing deformation is reflected in the large off-diagonal terms in the [A] matrix of an unbalanced laminate.

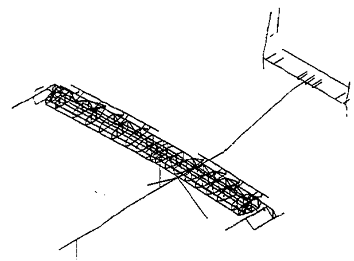
The matrix [B] in Fig. 6 relates twisting and warping of the laminate due to extension and bending. This twisting and warping can occur in laminates which are not symmetrical with respect to the centerline of the laminate. The warping tendency is an undesirable attribute from a manufacturing viewpoint, and can be avoided even in a tailored wing by simply stacking the plies symmetrically with respect to the centerline of the laminate.

Finite Element Models

The NASTRAN finite-element model shown in Fig. 7a is used for the tailoring study and consists of a built-up wing attached to a simplified elastic line model of the fuselage and empennage. This model has sufficient detail to model the tailored wing cross section, but is small enough to allow multiple design iterations. The rotor, pylon, controls, and drive system of this model are identical to the V-22 FSD model in Fig. 7b. Fig. 8 depicts the right wing of the tailored wing model and the cross-section geometry at midspan. The wing skin panels are modeled with CQUAD4 elements and the stringer and spar caps are modeled with CBAR elements. The wing design was originally a two-stringer configuration as shown, but later evolved into a three-stringer configuration during the design iterations. Initially, the stringer and spar cap areas in the tailored wing model were adjusted to match the stiffnesses of the five-stringer V-22 FSD model. As a check on the validity of this model, the frequencies and mode shapes of the fundamental modes important for proprotor stability were compared to the comprehensive V-22 FSD model and shown to be in good agreement.

Proprotor Stability Analysis

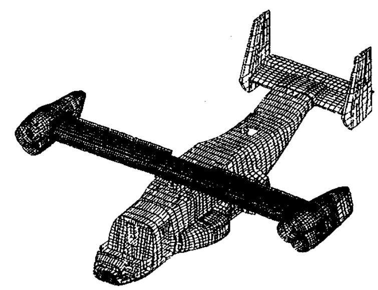
Wing/pylon/rotor stability is computed using the ASAP computer code developed by Bell. ASAP is an



a) Simplified wing and elastic line fuselage model

Tailored Wing Model

Grids	622
Elements	1,069
D.O.F.	2,269



b) V-22 detailed FSD model

V-22 FSD Stability Strength and Weight Reference

Grids	22,014
Elements	44,052
D.O.F.	123,086

Fig. 7. NASTRAN finite element models used in wing tailoring study.

eigenvalue formulation that can analyze proprotor stability for axial mode flight. The elements of the ASAP code include math models of the rotor, airframe, drive system, and control system, which are shown schematically in Fig. 9 and briefly described in the following paragraphs.

The ASAP rotor math model consists of rigid blades with discrete hinges for representing the flap, lag, and coning modes of the rotor, which have been found to be the principal modes involved in proprotor stability. The rotor aerodynamic model assumes uniform inflow with quasisteady aerodynamics. Detailed calculation of kinematic and elastic coupling parameters of the rotor are performed external to ASAP in more comprehensive rotor analysis codes and are input to ASAP. The airframe is modeled by elastic modes derived from the NASTRAN model. Mode shape information for the rotor hub and control plane is required for stability analysis. Structural damping inputs are based on measured values from ground vibration tests and wind tunnel stability tests. ASAP has achieved good correlation with measured stability boundaries from aeroelastic model tests, described in Ref. 2, and with full-scale V-22 flight test data, which establishes confidence in the capability to evaluate changes to the wing dynamics. ASAP generates the damping versus airspeed plots in Fig. 9, which show that the SWB and SWC modes are the critical modes of instability.

Loads and Stress Analysis Methods

The composite tailored wing loads and stress analysis is performed using a subset of the external loading conditions used for the V-22 design. The critical wing design loading conditions are used to evaluate the tailored wing design concepts for overall ply-by-ply strength, skin panel buckling stability, and wing stringer/skin strength and buckling stability. The analysis is performed using the same Bell production codes that were used on the V-22 FSD program. The stress analysis codes consist of (1) a laminate properties and strength analysis program, (2) a composite panel buckling analysis program, and (3) a composite wing segment analysis program.

STRUCTURAL TAILORING DESIGN ISSUES

This section investigates the fundamental design issues for a thin composite tailored wing. The

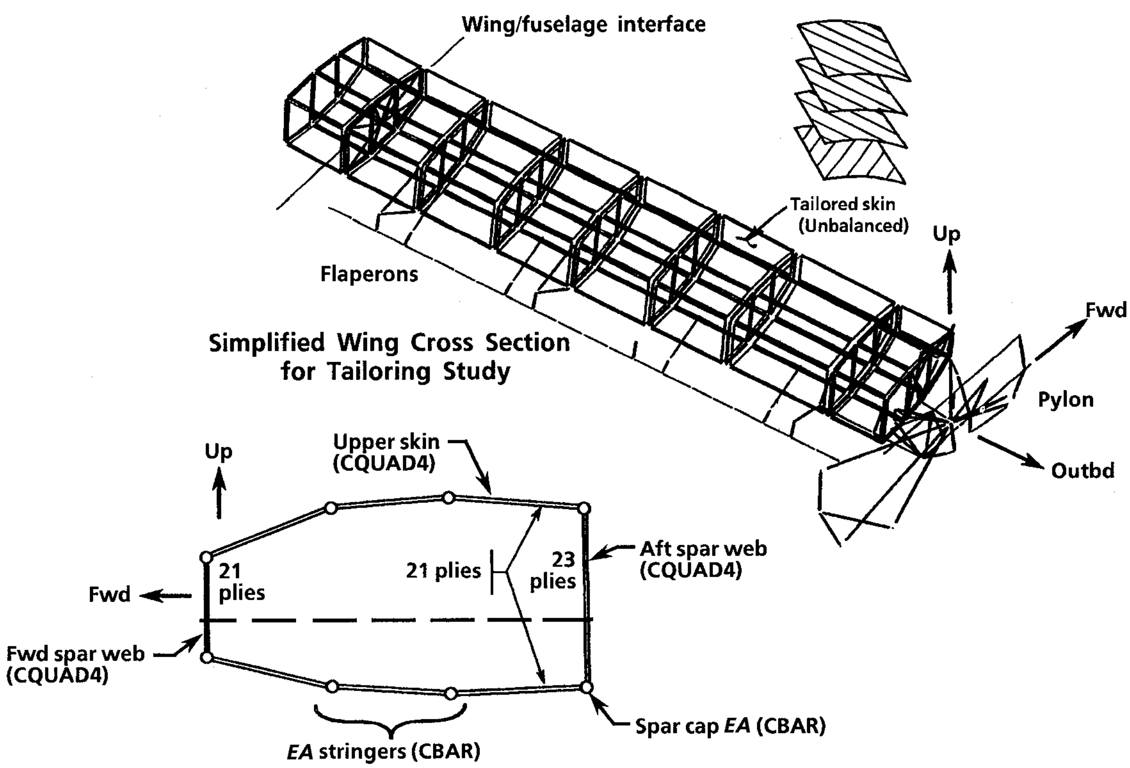


Fig. 8. NASTRAN model of wing used in tailoring study.

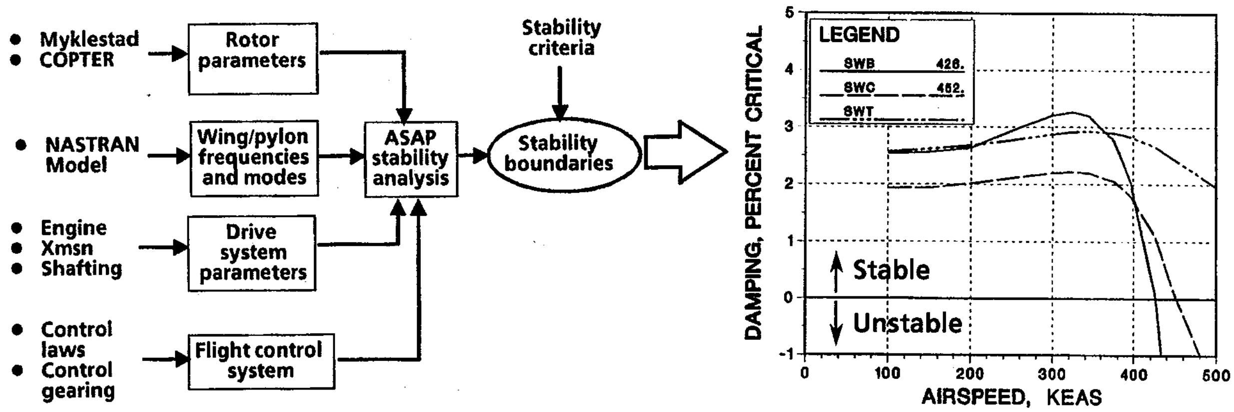


Fig. 9. ASAP proprotor stability analysis.

investigations include an evaluation of wing thickness (t/c) effects on the stability of untailored wings, and the use of a simple wing box model to study composite ply orientation effects on structural coupling and stiffness.

Effect of Wing Thickness on Stability

Wing thickness values of 23% t/c, 20% t/c, and 18% t/c are evaluated to quantify the magnitude of the wing thickness effect on stability, stiffness, and frequency placement of an untailored wing. The thickness variation is accomplished by simply changing the depth of the wing cross section, without changing the chord, cap area, or skin laminate from the baseline V-22 configuration.

Fig. 10 compares the SWB pitch/bending coupling, modal frequency placement, and stability boundaries for 23, 20, and 18% t/c wing configurations. As the wing thickness is reduced to 18% t/c, the SWB mode stability speed is reduced by 11 knots (kn), while the chord mode stability is reduced by 38 kn. The frequency of all three modes decreases as wing thickness is reduced, but the SWB mode changes by the largest percentage. Note also that the pitch/bending coupling of the SWB mode is improved as wing thickness is reduced. The relatively small reduction in SWB stability speed is attributed to the beneficial effects of pitch/bending coupling. The coupling is improved because of the increased frequency separation of the SWB and SWT modes, which tends to decouple wing beamwise bending and torsion. While the isolated effect of reduced beam frequency is detrimental to stability, the improvement in pitch/bending coupling for the 18% t/c wing partially offsets this effect, resulting in a small reduction in the stability boundary. The 38-kn decrease in SWC mode stability is directly related to the reduction in the SWC and SWT mode frequencies and to the detrimental change in the SWC mode shape. For the 18% wing, the SWC mode shape is more coupled with wing torsion, which reduces stability.

In order to achieve the same stability boundary as the 23% t/c baseline wing, composite tailoring must increase the stability of the 18% t/c wing SWB mode by at least 11 kn and the SWC mode by at least 12 kn.

Effect of Composite Ply Orientation on Pitch/Bending Coupling and Stiffness

The nomenclature used to describe the ply orientation is defined pictorially in Fig. 11. For convenience, a laminate “blend ratio” is defined which represents the ratio of -45-deg plies and +45-deg plies, since this ratio is a frequently varied parameter. For example, a laminate with 70% of the bias plies oriented at -45 deg and 30% of the bias plies at +45 deg is defined as a 70/30 blend. The blend ratio angles refer to the plies on the top surface of the wing skin panel when viewed looking down. An actual skin laminate with a 70/30 blend ratio is noted in Fig. 11.

To simplify the evaluation of skin ply orientation effects, the multi-bay rectangular wing box model shown in Fig. 12 is used. By applying static forces, F_X and F_Z, and moments, M_Y, to this model, the ply orientation configurations which yield maximum

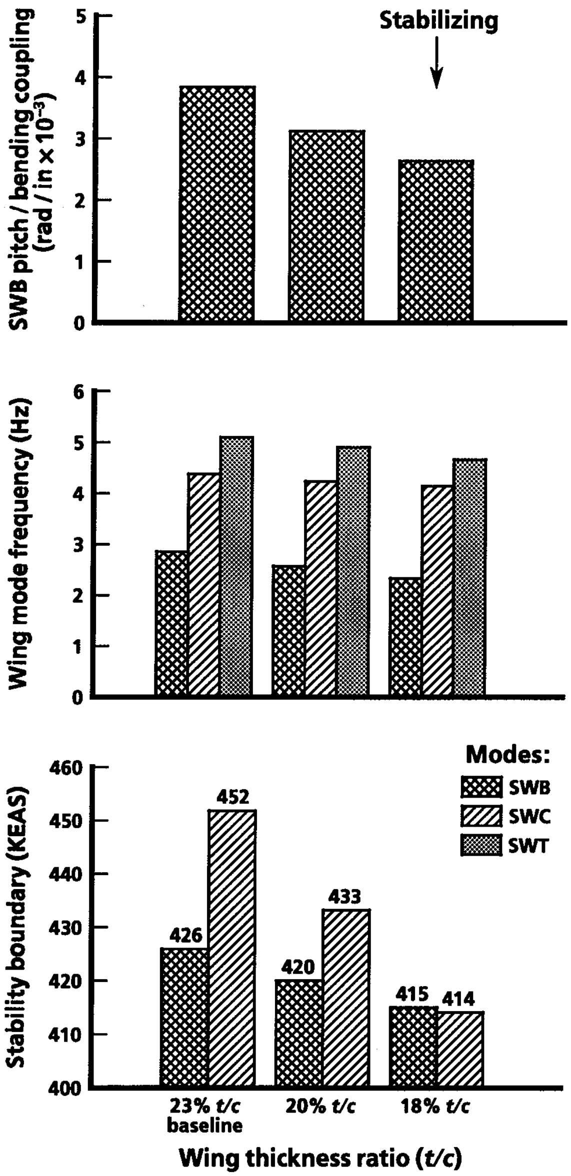


Fig. 10. Effect of wing thickness ratio on stability, frequency and pitch/bending coupling for untailored wings.

static pitch/bending coupling can be determined. First, the upper and lower skin ply orientations are varied while maintaining a balanced laminate for the forward and rear spar webs. This produces four possible configurations of skin ply orientation : (1) +45-deg upper skin (0/100 blend) with +45-deg lower skin (0/100 blend); (2) -45-deg upper skin (100/0 blend) with +45-deg lower skin (0/100 blend); (3) +45-deg upper skin (0/100 blend) with -45-deg lower skin (100/0 blend); and (4) -45-deg upper skin (100/0 blend) with -45-deg lower skin (100/0 blend). The configuration that produces the maximum beneficial coupling (nosedown pitch with up vertical deflection) consists of -45-deg plies for both the upper and lower skins. This arrangement has the upper and lower skin plies aligned from the inboard rear spar toward the outboard forward spar when viewed from above. For upward deflection of the wing, compression loads in the upper skin cause the skin to shear forward while tension loads in the lower skin cause the skins to shear rearward. Differential motion of the upper and lower skins induces the desired nosedown twist as shown in Fig. 12. Next, the best ply orientation for the forward and rear spar webs is determined by setting the upper and lower skin plies as defined above, and then examining the four possible combinations of ply orientations for the spar webs. This evaluation shows that unbalanced spar webs have a very small effect on pitch/bending coupling and cause an undesirable reduction in chordwise stiffness and strength. Based on the static analysis, the best choice for the forward and rear webs is a balanced laminate.

The effect of the upper and lower skin laminate blend ratio on the wing box pitch/bending coupling and static stiffness is shown in Fig. 13. Unit loads are applied to the rectangular wing box and the calculated deflections are used to compute stiffness and pitch/bending coupling. As laminate blend ratio is increased, all stiffness values are reduced, especially torsion. The beam and chord stiffness drops due to laminate tensile stiffness reduction, while the torsional stiffness drops rapidly due to the shear stiffness reduction described earlier. Although higher blend ratios improve pitch/bending coupling as shown in the figure, the loss in stiffness can be very detrimental to stability.

PARAMETRIC STUDIES OF TAILORED WING CONFIGURATIONS

In this section, parametric variations are performed on 23%, 20%, and 18% *t/c* wing designs to optimize the tailored wing for maximum stability at the lowest possible weight. The parametric trends for each of the three thickness ratios are similar. The parameter studies include (1) variations of both the skin and spar web laminate composition, (2) variations of only the skin laminate composition, and (3) combined skin tailoring and stringer and spar cap area tailoring.

Effect of Varying Skin and Spar Web Laminate Composition

The properties of the composite laminates comprising both the upper and lower skins and the forward and

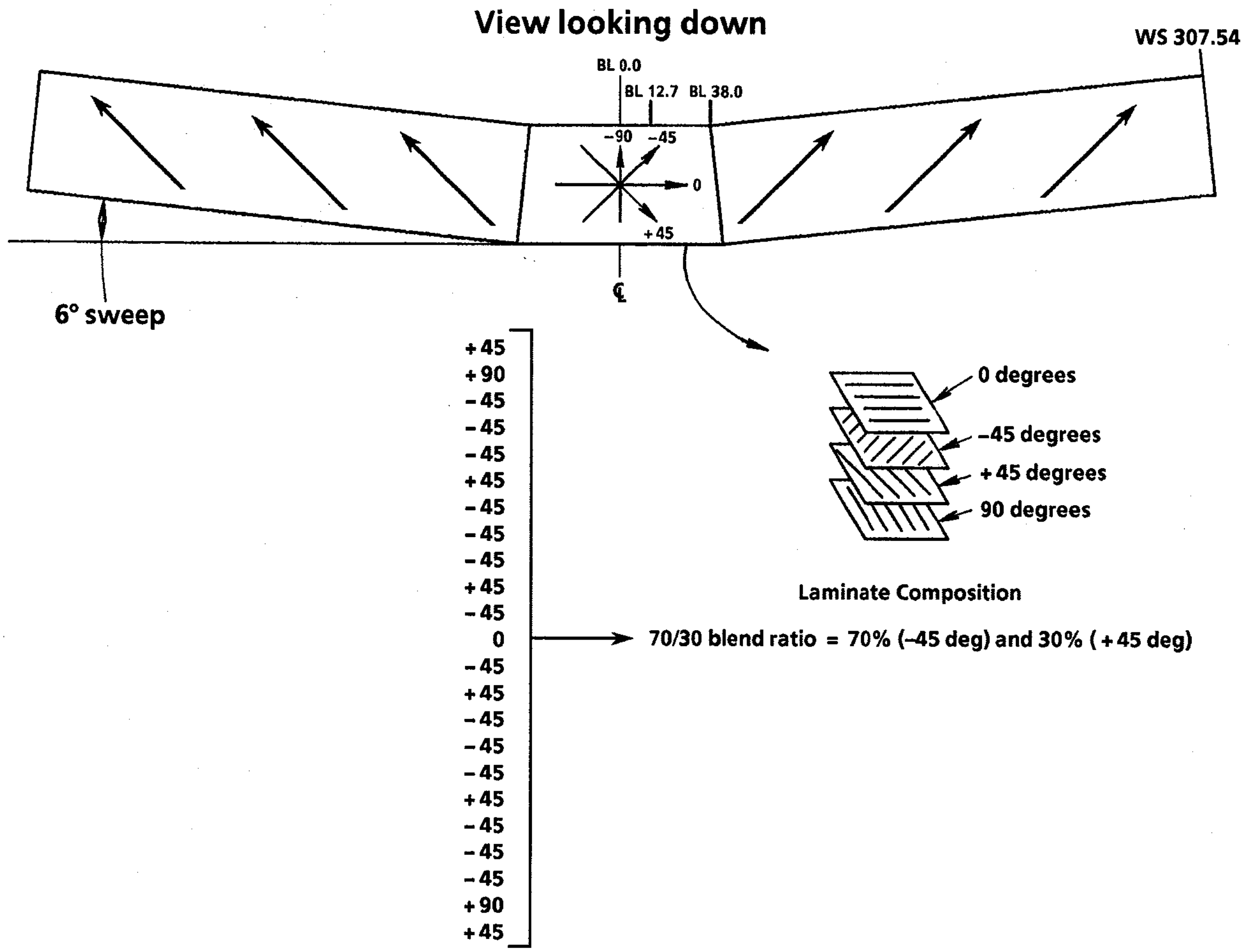


Fig. 11. Composite ply orientation nomenclature.

rear spar webs are varied uniformly to determine the effect on the stability boundary. In this study, the baseline V-22 wing laminate is modified by adjusting the blend of -45-deg and +45-deg plies in the laminate. The laminate blend ratio is varied incrementally from a balanced 50/50 blend ratio to a highly tailored 100/0 blend ratio. The summary data for the variations in blend ratio are presented in Fig. 14, which shows the pitch/bending coupling ratio of the symmetric wing beam mode, the frequency of the SWB, SWC, and SWT modes, and the stability boundary of the critical SWB and SWC modes.

The baseline V-22 23% *t*/*c* wing data are presented for comparison purposes. Case 1, in Fig. 14, represents an untailored 18% *t*/*c* wing, which has lower frequencies and less stability than the baseline 23% *t*/*c* configuration. As the laminate blend ratio is varied, the stability of the SWB mode reaches a maximum near a 70/30 or 80/20 blend ratio. Symmetric wing chord mode stability, however, is maximum with the 50/50 blend ratio and loses stability at higher blend ratios. Thus, conflicting demands are placed on the tailored wing design to achieve stability improvements for both modes.

The stability changes can be explained by examining the modal data trends in Fig. 14. For the SWB mode, the stability improvement for a 70/30 blend ratio is attributed to the favorable decrease in pitch/bending coupling and to the slight increase in wing beam frequency caused by a forward shift of the wing shear center. At blend ratios above 70/30, the principal cause of the stability reduction is the loss in torsional stiffness which is apparent from the torsional frequency drop. The loss in torsional stiffness is the most serious drawback of highly tailored laminates.

Unlike the SWB mode, the SWC mode stability is maximum at the 50/50 blend ratio and declines at

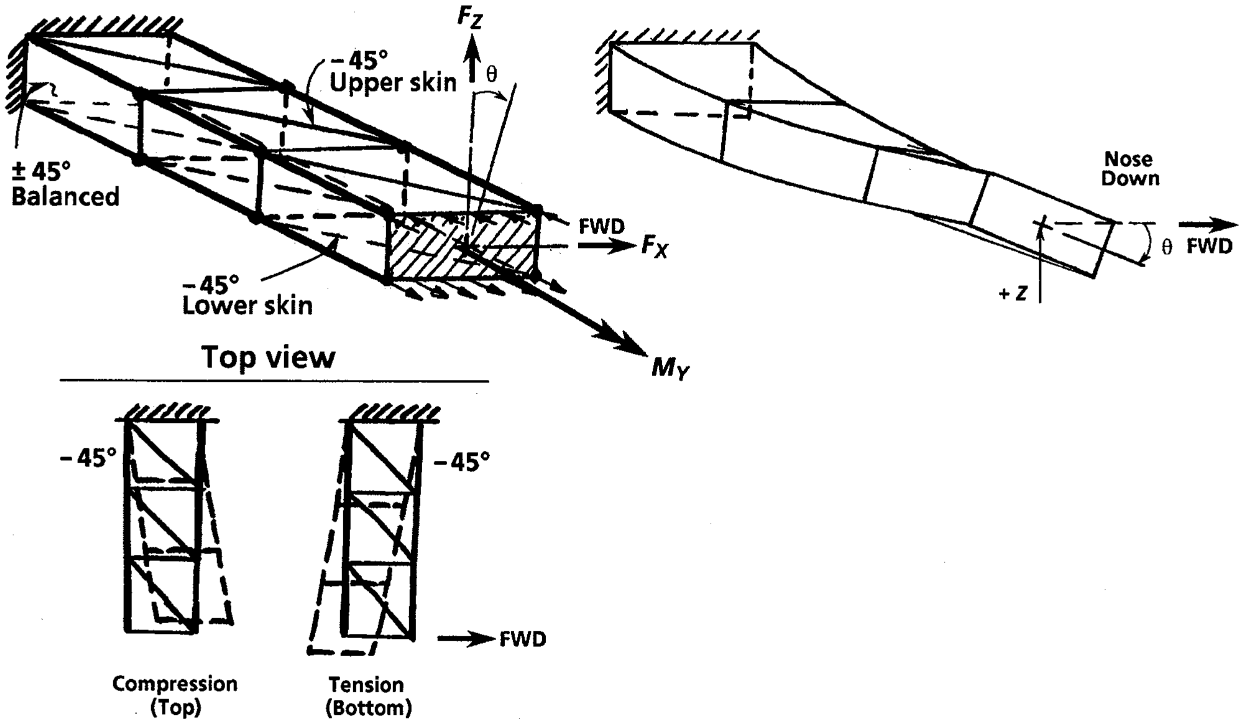


Fig. 12. Simple wing box model used to evaluate composite ply orientations.

higher blend ratios. This stability decline is caused by the reduction in chord and torsion mode stiffness and frequency at higher blend ratios. For highly blended laminates, the loss in chord stiffness can be traced to the presence of large off-diagonal coupling terms in the element extensional stiffness matrix [A] which effectively reduce the laminate stiffness. The SWC stability is also insensitive to the SWB pitch/bending coupling improvement at high blend ratios. Using high blend ratios for the forward and rear spar webs has a very small effect on the wing chord/torsion coupling, and this small coupling change has a negligible effect on the chord mode stability boundary.

Effect of Varying Only the Skin Laminate Composition

Consider the case of varying only the upper and lower skin laminates, while maintaining a 50/50 blend ratio for the forward and rear spar webs. Stability trends, as well as supporting modal data, are presented in Fig. 15 for varying only the skin laminate blend ratio. When compared to Fig. 14, the results in Fig. 15 show slightly higher stability speeds for all laminate blend ratios. This trend indicates that the spar web laminates should remain balanced in a composite tailored wing. This conclusion is supported by the earlier static stiffness results from the rectangular wing box which showed that highly tailored spar webs reduce wing stiffness without improving pitch/bending coupling.

In Fig. 15, the SWB mode stability reaches a plateau from the 70/30 to 100/0 blend ratios. Between the 70/30 and 100/0 blend ratios, the beneficial effects of pitch/bending coupling offsets the detrimental effects of reduced torsional stiffness to maintain essentially the same SWB stability boundary. Fig. 15 also shows that the SWC mode stability decreases significantly with increased wing skin blend ratio in a manner similar to the data in Fig. 14. The SWC mode stability is somewhat higher when tailoring only the skin laminate because the forward and rear spar webs are stiffer and favorably increase the chord mode frequency to raise the stability boundary.

Effect of Stringer and Spar Cap Area Tailoring

Skin laminates with high blend ratios decrease the stability of the SWC mode due to reductions in wing chordwise stiffness and frequency placement. This section examines the influence of tailoring the cap

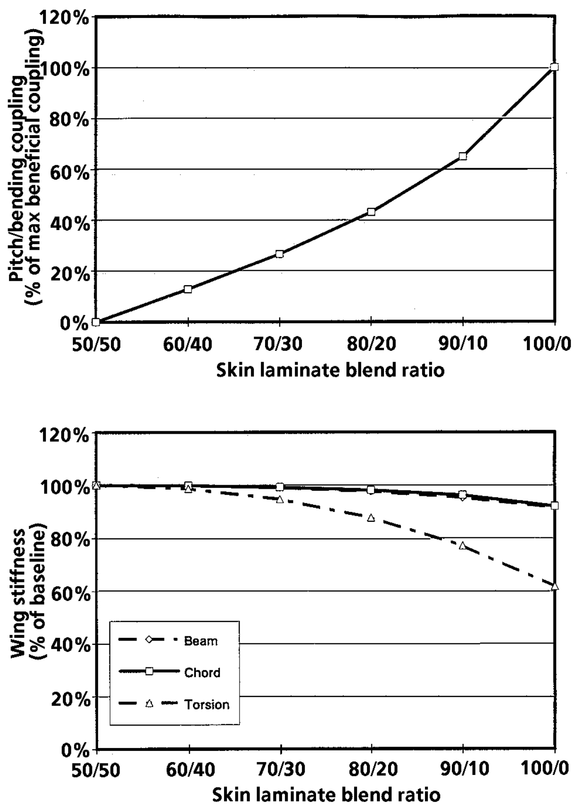


Fig. 13. Effect of laminate blend ratio on stiffness and static pitch/bending coupling.

area (*EA*) of the stringers and spars to compensate for the loss in stiffness due to skin tailoring. The stiffness increase can be accommodated by either redistributing portions of the stringer cap area to the spars or by directly increasing the area of the forward and rear spar caps.

The effects of transferring stringer area to the spar caps for a 20% *t*/*c* wing is illustrated in Fig. 16. All of the wing configurations in this survey have an 80/20 skin laminate blend ratio which maximizes the SWB mode stability. For this parameter study, up to 50% of the stringer cap area is removed from the stringers and transferred to the forward and rear spar caps as shown in Cases 1–5. A simple stress analysis shows that transferring more than 50% of the stringer cap area causes strength problems and therefore is not investigated.

Transferring some of the stringer cap area to the spar caps is beneficial to both the SWC and SWB stability boundaries. An explanation for the stability improvements is contained in the supporting modal data in Fig. 16. As the cap area is transferred from the

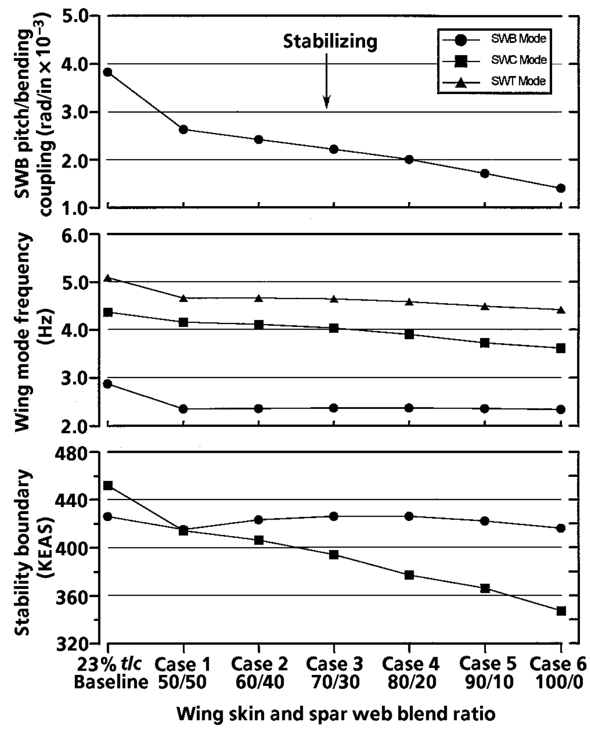


Fig. 14. Effect of skin and spar web blend ratio for an 18% *t*/*c* wing

stringers to the spar caps, the wing beamwise stiffness and frequency are lowered due to reduced effectiveness of the cap area (Ad^2) since the stringers have a larger "*d*" than the forward and rear spar caps. Concurrently, the increased area added to the forward and rear spar caps raises the chord stiffness and frequency. Although the cap area transfer has no direct effect on the torsional stiffness, the torsion frequency is nevertheless increased due to chord/torsion coupling as the chord mode is moved closer to the torsion mode. The reduced beam frequency and the increased torsion frequency increases the frequency separation of these two modes which has a favorable effect on pitch/bending coupling and results in the 8-kn improvement in the beam mode stability speed. The SWC mode stability benefits by 6 kn for 50% cap area transfer due to increased chordwise stiffness and frequency. The transfer of stringer cap area to the forward and rear spar caps provides these stability benefits with no net increase in weight. For further improvements to the SWC mode stability boundary, the most weight efficient method is to add cap area directly to the spars.

LOADS AND STRESS ANALYSIS

Promising tailored wing configurations are examined for strength with the same analysis codes used on the

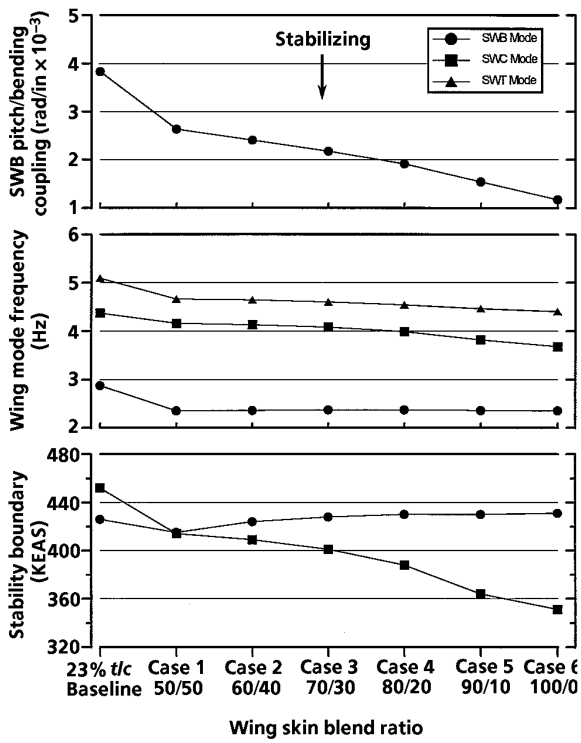


Fig. 15. Effect of wing skin blend ratio for an 18% *t/c* wing.

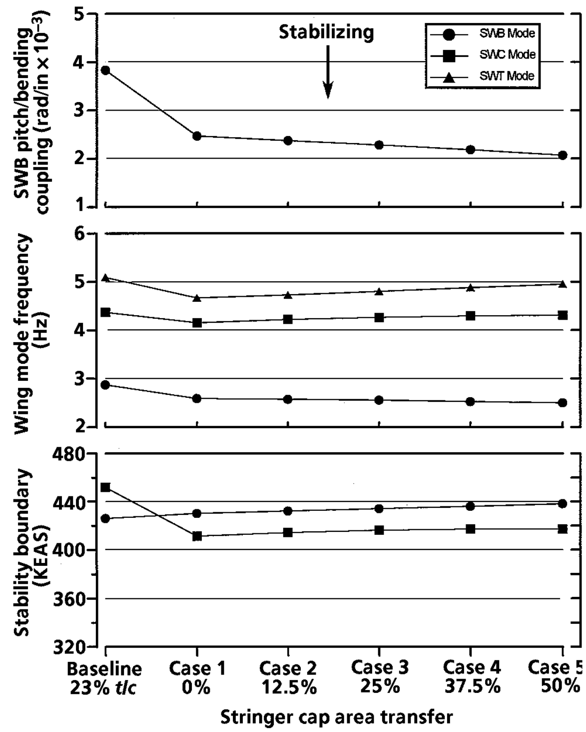


Fig. 16. Effect of transferring stringer cap area to the spar caps for a 20% *t/c* wing with an 80/20 skin blend ratio.

V-22, but with a reduced set of critical loading conditions. From the complete set of V-22 design load conditions, a subset of four critical load conditions for the wing root are used in the stress analysis to verify the structural strength of the composite tailored wing. The stress analysis examines the most critical inboard wing station for skin panel buckling, ply strength, and wing stringer/skin buckling.

Initially, an 18% *t/c*, two-stringer tailored wing configuration was developed by using the results of the parametric tailoring studies to select a combination of skin, stringer, and spar cap tailoring to satisfy the stability requirements. Stress analysis of the tailored two-stringer configuration, however, revealed two structural deficiencies. First, the two-stringer arrangement shown in Fig. 8 provides insufficient support for the tailored skins to prevent skin buckling. A three-stringer configuration is required to reduce the size of the skin panels to prevent buckling. Second, two additional skin plies are required to increase torsional strength to eliminate negative margins of safety in the skin.

FINAL COMPOSITE TAILORED WING CONFIGURATION

A final 18% *t/c* tailored wing configuration is defined using the results of the tailoring parameter studies and the findings from the stress analysis as a guide to optimize the configuration. The results of the parametric studies can be summarized as follows. First, the upper and lower wing skins should be tailored but the forward and rear spar webs should be balanced laminates. Second, the SWB mode stability is maximized with a 70/30 or 80/20 laminate blend ratio for the upper and lower skins, while SWC mode stability is most stable with a balanced 50/50 laminate blend. Third, transferring or adding cap area to the forward and rear spar caps is required to improve the chord mode stability. Finally, the stress analysis revealed that a three-stringer configuration was necessary, and that two additional skin plies are required to eliminate negative margins of safety for the composite tailored wing.

The final tailored wing configuration, depicted in Fig. 17, is an 18% *t/c*, three-stringer configuration with a

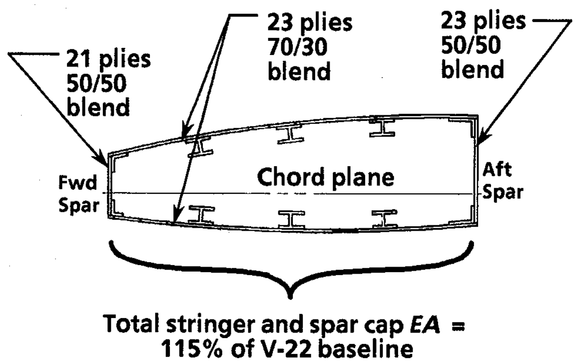


Fig. 17. Final 18% *t/c* three-stringer tailored wing configuration.

70/30 blend ratio for the skin laminate and a 50/50 blend ratio for the spar web laminate. The total stringer plus spar cap area is increased by 15% over the baseline 23% *t/c* wing to satisfy chord mode stability requirements. Relative to the baseline 23% wing, the total stringer cap area has been reduced by 40% while the total spar cap area has been increased by 160%. Due to strength considerations, the tailored wing also has two more additional wing skin plies than the baseline 23% *t/c* wing.

Weight estimates are made for the tailored wing configuration by computing the change in weight due to the addition and removal of structure from the baseline configuration. The skin ply and spar cap area changes increase the weight; however, the increase is almost offset by the weight saving due to the reduction in overall wing thickness from 23% *t/c* to 18% *t/c*, resulting in only a 29-lb (1.2%) increase in overall wing weight.

Fig. 18 compares the stability boundaries of the baseline 23% *t/c* untailored wing to the tailored wing configurations, and shows the incremental changes in stability as the wing is progressively tailored. Case 1 is structurally identical to the baseline configuration except the wing thickness ratio has been reduced to 18%. This configuration has a stability boundary which is 12 kn below the baseline due to the stiffness losses associated with the thickness reduction. The stability loss in Case 1 is recovered through tailoring as shown in Cases 2 and 3. Case 2 shows that the stability boundary can be increased 17 kn by spar cap area tailoring alone. Adjusting the skin blend ratio from 50/50 to the optimum 70/30 blend ratio provides a net 7-kn increase in stability boundary, as shown in Case 3, which is the final tailored wing configuration. Case 3 represents a 24-kn

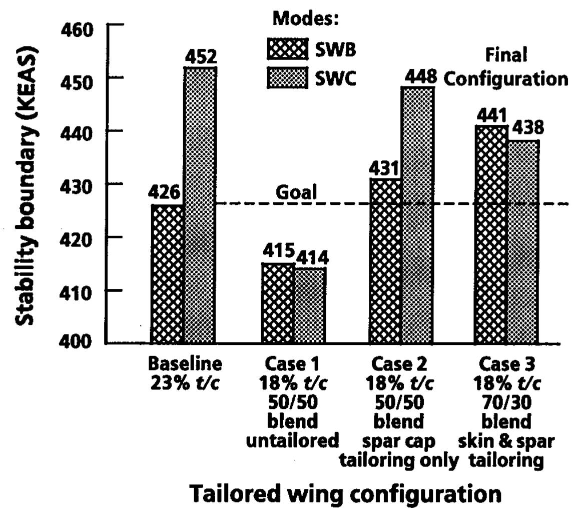


Fig. 18. Tailored wing stability summary.

improvement in stability boundary over the untailored wing in Case 1. In addition, Case 3 has a stability boundary which is 12 kn higher than the baseline 23% *t/c* wing, yet the weight penalty is only 29 lb out of a total wing weight of 2500 lb.

The benefit of composite tailoring is likely to be configuration dependent and other tiltrotor wing/pylon arrangements may yield more or less benefit from the tailoring schemes explored in this study. For example, changes to wing span and chord, pylon mass properties, and rotor weight can affect the wing frequency placement and modal coupling, and therefore influence the tailoring approach.

SUMMARY AND CONCLUSIONS

The results of this feasibility study, using the V-22 as the baseline configuration, show that proprotor stability gains are possible using composite tailoring to modify the dynamic characteristics of the wing. However, the overall stability gains available from composite tailoring are reduced by the conflicting structural tailoring requirements to satisfy both the beam and chord mode stability requirements. Parametric studies identified several important trends for the specific wing configuration examined here.

1. The forward and rear spar webs should use a balanced 50/50 blend ratio for maximum stability.

2. The stability of the SWB mode is maximum with a 70/30 or 80/20 skin laminate blend ratio, while the SWC stability is highest with a balanced 50/50 blend ratio.

3. Transferring stringer cap area to the forward and rear spar caps is beneficial for both SWB and SWC stability boundaries.

The 18% *t*/*c* composite tailored wing developed in this study has the following features:

1. The composite tailored wing exceeds the stability of an 18% *t*/*c* untailored wing by 24 kn. Stringer and spar cap tailoring provides 71% (17 kn) of the stability improvement, while composite skin tailoring accounts for 29% (7 kn) of the total 24-kn stability increase.

2. The 18% *t*/*c* composite tailored wing exceeds the stability boundary of the 23% *t*/*c* baseline wing by 12 kn and meets the stability requirements for a proposed civil tiltrotor.

3. The 18% *t*/*c* tailored wing is feasible from the stress and weight perspectives. Margins of safety based on an evaluation of previously determined critical load conditions are acceptable for the root of the wing. The 18% *t*/*c* tailored wing represents a weight penalty of less than 30 lb (1.2% of the wing weight) relative to the baseline 23% untailored wing.

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